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No. 718

DEVELOPMENT OF THE RULES GOVERNING THE STRENGTH OF AIRPLANES

By H. G. Küssner and Karl Thalau

PART III

LOADING CONDITIONS IN FRANCE, ITALY, HOLLAND AND RUSSIA - AIMS AT STANDARDIZATION

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PART III. LOADING CONDITIONS IN FRANCE, ITALY, HOLLAND AND RUSSIA - AIMS AT STANDARDIZATION

8. French Loading Conditions

The French strength specifications originally avoided all numerical data and left the test decision open for each individual case. Aroused by the rapid development of pursuit and acrobatic airplanes toward the end of the war, the assumption of sudden pull-out at high angle of attack from a vertical nose dive, formed the basis upon which to analyze wing strength. On the premises that the drag coefficient of the whole airplane in a nose dive is approximately equal to the drag coefficient c_{w_h} for maximum horizontal flight v_h at ground level, the S.T.Aé. prescribed the classical formula

$$n_A = k \frac{F}{N} (0.036\ v_h)^3 = 0.007 \frac{k\ \eta}{\rho_o\ c_{w_h}} \qquad (50)$$

for the load factor of case A.

The center of pressure was at one third of the wing chord.

In 1922 (reference 61) the k factors were prescribed:

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*"Die Entwicklung der Festigkeitsvorschriften für Flugzeuge von den Anfängen der Flugtechnik bis zur Gegenwart." Luftfahrtforschung, June 21, 1932, pp. 38-52. (For Parts I and II, see N.A.C.A. Technical Memorandums Nos. 716 and 717.)

Pursuit single-seat monoplane,	$k =$ 15	$(m^4/kg\ s^2)$
" " multiplane,	10	
Other military monoplanes,	11	
" " multiplanes,	7.5	
Nonmilitary monoplanes,	9	
" multiplanes	7.5	

The inclination of the resultant of the air loads toward the wing chord shall be 4:1 in case B. The point of application shall be determined from the wing polars. In case C the stress of the wings is investigated by its internal drag. The load factor in cases B and C is a stated fraction of the A-case load factor. In case D the load factor shall be $n_D = 0.5\ n_A$ for all airplanes.

Niles, after critically comparing formula (50) with the U. S. load factor, came to the conclusion that according to it some of the newer pursuit airplanes would be of inferior strength, whereas commercial airplanes, which practically never get into a nose dive, would become excessively strong.

Breguet and Devillers (reference 63) also criticized this formula and adduced the empirical breaking load factor, especially for commercial airplanes, from the stress in a vertical gust. Starting from the reasoning that a sharp pull-out at high speed is an unduly vitiating loading condition for commercial types, and that such a maneuver was not at all executable, particularly with large airplanes, they analyzed the motion of an airplane flying into a gust roller with sinusoidal distribution of the vertical velocity under the assumption of steady lift coefficients. The maximum stress is reached in the case of sudden rise of vertical velocity

$$n = 1 + \frac{\rho}{2} \frac{d\ c_a}{d\ \alpha} v_h\ w \frac{F}{G} \tag{51}$$

With

$$\frac{\rho}{2} \frac{d\ c_a}{d\ \alpha} \sim 0.25\ kg\ s^2/m^4$$

gust velocity $w = 3$ m/s, safety factor 2.5 of static quota and 5 as that of the dynamic quota, the breaking load factor becomes

$$n_{Br} = 2.5 + 3.75 \frac{v_h F}{G} \tag{52}$$

The breaking load factor by this formula deviates for different commercial airplanes only slightly from 6 as contrasted with formula (50) which, even with the minimum $k = 7.5$ yields unnecessarily high load factors in some cases. It was therefore believed that a constant load factor of 6 was perfectly sound for commercial aircraft. But this no longer holds true to-day, where the number of airplane types has increased consistently.

The Permanent Commission for Aeronautical Research, with which the S.T.Aé. and the International Commission for Air Navigation were affiliated, came to the conclusion in 1925 (reference 64) that formula (50) rendered the preliminary static analysis difficult, because the speed v_h was determinable only after test flights. Moreover, since the arbitrary k factors were simply empirical, a determination of the load factors independent of the speed but dependent upon the gross weight of the airplane, was preferable.

The load factors set up by the two Commissions are given in

Table XXIX. French Load Factors, 1925

		Total weight G (t)	n_A $\leqslant 1$	n_A $\geqslant 5$	n_C
Civil	CINA	Normal purposes	7	5	1.5
		Records and special purposes	5	4	1.2
		Acrobatics	9	7	2.5
	S.T.Aé.	Normal purposes	8	6	2
		Records and special purposes	6	6	1.5
		Acrobatics	12	9	3
Military	S.T.Aé.	Heavy bombers, training and ambulance airplanes	8	6	2
		Multiseaters, day bombers	9	7	3
		Pursuit and observation airplanes	13	10	4

(t × 2204.62 = lb.)

These load factors were based in part upon acceleration measurements made by Huguenard, Magnan, and Planiol (referenĉe 65).

For case B (c.p. position corresponding to that for maximum horizontal flight), the load factor is $n_B = 0.75 n_A$.

Case C shall be analyzed for a nose dive with terminal velocity; the load factor, better called safety factor in this case, is given in table XXIX.

For wheel landing from normal flight attitude (pancaking), the impact factor is 6 for all airplanes except for the special group, where it shall be 4.5.

As vertical component of the landing shock for the landing gear 5 times the gross weight of the airplane shall be assumed (3.5 times for special group). The resultant slopes 27° forward and 9° sidewise against the vertical. For the rest, the specifications were similar to those found in the 1927 edition of the Bureau Veritas.

The CINA, originated in France, began in 1925 with the promulgation of "minimum requirements for obtaining an airworthiness certificate." The loading conditions contained therein had, in May 1929, progressed to the following stage:

General Specifications for Stress Analysis and Testing

The tests or stress analyses are subject to the following rules:

a) For the successively assumed flight attitudes or movements on the ground the loads produced under these conditions and which the different parts of the airplanes have to carry, are determined and, except for the forces set up by the propeller, multiplied by the load factor cited in the subsequent chapter.

b) The forces produced by the propeller are introduced in actual magnitude when computing the airplane speed. In case of fatigue stress of the airplane, thrust and propeller torque shall be multiplied by the load factors given in the subsequent section if these load factors are less than 2.5; but in any other case, with 2.5.

When static strength tests are required, it must be proved during these tests whether the total stress assumed according to the above data, produces forces which actually cause failure in some part of the structure.

Granted sufficient design data, they may be referred to breaking limit or elastic limit; but in all cases the different assumed load factors must be such as to give assurance of an identical factor of safety as the static strength tests with the load factors (given in the next section) would reveal.

Analysis and Strength Test of Wings

Case I: Flight with c.p. far forward. This case corresponds to pull-out from a nose dive and to horizontal flight in a vertical up-gust.

It shall be assumed that the airplane flies horizontally at the angle at which the center of pressure of the air loads is farthest forward. The forces impressed hereby on the different parts of the airplane shall be analyzed and the following breaking load factors applied thereto:

Gross weight of airplane	< 1 t	1 to 5 t	> 5 t
Class 1 (normal)	7	7 " 5	5
" 2 (special)	5	5 " 4	4
" 3 (acrobatic)	9	9 " 7	7

The load factors for airplanes having a total weight of from 1 to 5 t change linearly.

Case II: Flight at maximum speed.

The airplane shall be assumed to fly horizontally at its top speed v_h without the power and r.p.m. of the engines exceeding their respective internationally accepted figures. The loads impressed thereby on the individual parts of the airplane shall be analyzed and the pertinent load factors applied; they are three fourths of the value of case I.

Case III: Nose dive (c.p. farthest to the rear).

The airplane shall be assumed to dive at its limiting velocity with power off. The loads impressed hereby upon the individual parts of the airplane shall be analyzed and the following load factors applied:

Airplanes of class 1 (normal)	1.5
" " " 2 (special)	1.2
" " " 3 (acrobatic)	2.5

Case IV: Rough landing.

The airplane shall be assumed to be in horizontal attitude and drop vertically when touching the ground, after which the weight of the different members of the structure shall be multiplied as follows:

Airplanes of class 1 (normal)	6
" " " 2 (special)	4.5
" " " 3 (acrobatic)	6

Aside from the four main cases, there are the following special cases:

a) It shall be assumed that the airplane attains to attitudes 1 and 2 successively; hereby half of the above-cited load factors for analyzing the produced forces are assumed, and it must be proved whether, after failure of one bracing or fitting, any part of the cellule is under greater than its breaking load.

b) The loads on the wings shall be analyzed for the case that the airplane taxies or that the engines rotate on the ground separately or collectively, whereby the highest permissible torque shall be assumed and a unit load factor of 2.5 applied.

Analysis and Test of Control Surfaces

The vertical tail surfaces of the airplanes of class 1 (normal) and of class 2 (special) shall be designed to withstand a mean test load perpendicular to their surfaces, which is defined according to the formula $Q = 3.6\ v_h$, but which in no case must be less than 70 kg/m².

The distribution of this mean load over the fin surface shall be uniform, triangular over the rudder. The apex of the triangle shall lie over the outer edge, its base over the axis in unbalanced, and over the leading edge in balanced, elevators.

The strength of the fin attachment to the fuselage and of the rudder must be at least equal to the applied loads.

Fin and rudder and fittings must, without abnormal fatigue, sustain the stresses set up by control forces in flight or on the ground.

These regulations were revised July 1931, and amended as follows:

Elevators and stabilizers shall be analyzed with that of the following loads which produces the greatest stress:

a) A steady load equal to that specified for the vertical tail surfaces.

b) The loads resulting from the equilibrium equations for the first three flight attitudes with the same load factors as for the wings.

c) The load set up when the part of the elevator lying on one side of the line of symmetry of the airplane is loaded separately. If not amenable to direct analysis it may be assumed that the corresponding load is for the time being half of the loads found under a) and b).

For the investigation of the equilibrium equations in case b), the c.g. of the airplane yielding the maximum load on the control surfaces shall be assumed.

The load distribution over top and bottom of wing on one hand, and over span and chord on the other, depends upon the results from full-scale or model tests. When such are not available, officially recognized publications may be consulted. The fin attachments on the fuselage and of the elevator must be designed to withstand at least the stresses produced by the loads on the tail surfaces.

Ailerons shall be analyzed for the loads accruing from the second load case, the ailerons shall be assumed to be displaced 3° downward, and the load factors for the wings shall be applied; intensity and distribution of the loads to be taken from experiments or, lacking these, from officially recognized publications. Aileron fittings shall be designed to withstand at least the stresses impressed by the aileron loadings.

Landing Gears

For landing-gear design, three conditions must be complied with:

1. With an airplane in flight attitude, it shall be assumed that only the wheels touch the ground. The total weight shall be multiplied by a load factor 4.

2. The airplane attitude is as above, but the resultant of the loads is no longer vertical but shall be assumed inclined in a plane perpendicular to the longitudinal axis of the airplane so that the horizontal component equals 0.7 times the gross weight of the airplane.

3. The airplane is in the same attitude and subjected to the same loads as in the first case. But the resultant of the loads shall be assumed to be inclined in a vertical plane through the longitudinal axis of the airplane, so that the horizontal component is equal to one fourth of the resultant.

The stresses of the parts supporting the fuselage shall be analyzed as follows:

a) The airplane rests with the wheels and the support on a horizontal plane and its weight shall be multiplied with the load factor set up for the landing gear.

b) If a skid is used, it and its fitting must be designed for 50 percent of the preceding vertical loads, and for any position which the skid may assume.

It must be proved that the loads on the landing gear produced when the airplane in flight attitude drops vertically from a height h above the ground, do not exceed the above cited loads. The height h shall be 20 cm for airplanes having a landing speed not exceeding 65 km/h. For those with a landing speed of more than 65 km/h, 1 cm shall be added for each kilometer. When wheel brakes are used the braking forces shall not produce abnormal wear.

Fuselage, Engine Frame, and Various Accessories

Fuselage, engine supports, and their accessories (particularly cockpits, tanks, and their mountings), shall be analyzed for stresses in taxying and in flight. It shall be assumed:

a) that wings, control surfaces, landing gear, and other parts supporting the fuselage are loaded according to the load schedule;

b) that these loads are used to determine the inertia reaction of the static and dynamic loads of the airplane;

c) that the loads and reactions cited under a) and b) occur simultaneously and under identical conditions;

d) that the forces produced by the propeller are multiplied by the load factor which corresponds to the pertinent airplane motion.

In multi-engine airplanes the unsymmetrical forces which may occur when one or more engines stop, must be taken into account. And lastly, the operating and transmitting parts of the control systems as well as their fittings shall at least be equal to the forces produced by the loads assumed for the elevators.

Because of the international aspect of these regulations a comparison with the German loading conditions should be of interest.

Case 1 of the CINA corresponds to case A and is most important for the strength of airplane wings. For smaller commercial airplanes of from 2 to 4 t weight, the CINA specifies about 40 percent greater strength than German airplanes which have proved their worth in many thousands of flying hours. Sport and light airplanes also must, according to the CINA specifications, have a 35 percent greater strength than that of reliable German types. If the CINA had stipulated a higher average speed range, the higher load factor would admittedly be justified but not for airplanes having a low speed range. Several French experts likewise have raised objections to the load factor of case 1.

Case 2 differs from case B insofar as the actual lift coefficient is arbitrarily deduced from maximum horizontal flight, although the required high-load factor of this case $n_2 = \frac{3}{4} n_1$ is only attainable at the terminal velocity of the nose dive, and even then is not attainable with airplanes having a high load factor (acrobatic group). In other words, the CINA specifies especially high safety for pull-out from high gliding and diving speed, which is not customary with commercial and sport airplanes. The same applies to case 3, which corresponds to our C case. The CINA specifies for all airplanes, not only for the acrobatic group, the nose-diving condition with terminal velocity. This requirement is contrary to practical experience, according to which diving with terminal velocity is practically nonexistent even with larger airplanes. The German as well as the English strength specifications provide therefore a conformably lower C-case speed.

The formula for control surface stress corresponds, relative to the linear relationship with the maximum horizontal speed, to the gust formula (equation 30) of the DLA.

The aileron loading is only about half of that prescribed by the DLA. It is questionable whether the ailerons should be designed stronger even for reasons of rigidity and gripping strength.

In 1927 the specifications of the Bureau Veritas were as follows:

Case C: dive with terminal velocity.

The propeller acts as windmill with 15 percent higher r.p.m. The propeller drag shall equal the wing drag for small airplanes, and less for large airplanes.

Case B: horizontal flight at maximum speed, corresponding to pull-out from a nose dive.

Case A: center of pressure far forward, corresponding to maximum angle of attack reached at end of nose dive.

The required breaking-load factors are tabulated in

Table XXX. Load Factors, Bureau Veritas, 1927

	Normal			Aerobatic		
G (t)	< 1	1 to 5	> 5	< 1	1 to 5	> 5
Case A	8	8 to 6	6	12	12 to 9	9
" B	6	6 " 4.5	4.5	9	9 " 6.75	6.75
" C	1.5	1.5	1.5	2	2	2

In airplanes designed for inverted flight, the corresponding loads in opposite directions shall be assumed at three fourths of the above cited load factors.

The wing cellule shall be designed to withstand asymmetrical stresses equivalent to the maximum aileron loads multiplied by the above load factors. If there is uncertainty regarding this, the load factor on one wing-half shall be equal to n; on the other, equal to n - 1.

The landing gear shall be designed to withstand a static load of five times the gross weight of the airplane, as well as a free drop from 0.5 m height for day airplanes and from 1.0 m height for night airplanes. The remainder of the airplane must be able to withstand the impact factor 6 in landing. The landing gear shall also be strong enough to sustain oblique forces sloping at 9° against the plane of symmetry and one-wheel landing.

Wings and fuselage shall be designed for higher fatigue stress than that for which the landing gear is analyzed.

The airplane shall have a safety factor of 2.5 by static propeller thrust and torque. Ailerons shall be designed to carry a load of 200 kg/m², and the control surfaces for 60 percent of the maximum wing loading. The control-operating system shall be designed to withstand the maximum elevator loads as well as frictional and impact loads.

To avoid forced wing oscillations (while in ignorance of the complicated processes), airfoils with small c.p. displacement are recommended. It was probably believed that the center of gravity of the wings would thereby be shifted farther forward, and so increase the critical speed.

In 1929 the Bureau Veritas published a revised load schedule, plainly patterned after the loading conditions of the D.V.L., which had been published in the meantime. The excess stress due to temperature variation within the anticipated limits shall be investigated.

All parts must be designed to carry the maximum static loads with a factor of safety of 2. In addition, it must be proved that all parts can carry the 1.5 times static loads without distortion; that is, the elasticity (yield) limit of the material must not be exceeded under the loading. In metal parts, subject to considerable vibrations, the stress in normal flight must not exceed the fatigue limit.

The choice of calculation method is left free, but it is recommended to make the static analysis with the breaking load equal factor of safety times outside load.

Aside from some special cases, the maximum stresses occur in pull-out from the greatest prescribed nose dive (emergency position) and are greater as the diving speed is greater and the more suddenly the pull-out is effected.

For flight with maximum lift the factors of the gravity forces (safe load factors) given in table XXXI shall be used as outside loads.

This maximum lift shall be assumed once for maximum angle of attack (lift coefficient), then for maximum diving speed. Magnitude and direction of the resultants of the air loads shall be taken from the polars.

Table XXXI. Safe Load Factors, Bureau Veritas, 1929

Gross weight of airplane (t)	< 1	1 to 5	> 5
Normal	3.5	3.5 to 2.5	2.5
Acrobatic	7.0	7.0 " 6.0	6.0
Acrobatic with speed limitation $v_c \leq 1.3\ v_h$	4.5	4.5 " 3.5	3.5

a) When the nose dive is not limited by any distinct reaction of the elevator, the terminal velocity with allowance for propeller and elevator drag shall be used as diving speed.

b) When a reaction of definite magnitude on the elevator (10 to 15 kg for $G < 5$ t, 20 to 30 kg for $G > 5$ t) is not exceeded by the pilot, then the maximum steady gliding speed belonging to the corresponding elevator moment shall be considered as superior speed limit.

In any case it should at least be equal to 1.25 times the maximum horizontal speed (full throttle).

The airplane shall have a factor of safety of 2 for stresses at this speed (lift from 0 to maximum).

The stress of an airplane in horizontal flight (cruising speed) or in gliding with power off upon entry from calm air into a vertical air current (gust) of $\pm$ 11 m/s velocity shall be investigated. Factor of safety, 2.

For large airplanes the local overloading by gusts whose intensity is about 1 to 2 times that of steady lift, shall be investigated.

Normal airplanes shall be designed for downward pressure (inverted flight) with 50 percent (acrobatic group, 75 percent) of the load factor given in table XXXI. Airplanes designed for diving with terminal velocity shall be strong enough to withstand the stresses in the range of the angles of attack contiguous to this flight attitude.

With a view to unsymmetrical stresses the stress of one wing-half in normal airplanes shall be reduced by 1.0 times; in acrobatic airplanes by 2 times the steady lift.

In case of failure of one bracing member, the remainder must at least be strong enough to sustain 50 percent of the specified loads. All loading conditions shall be in combination with additional stresses due to maximum elevator displacements and 2.5 times the propeller thrust and torque in normal horizontal flight.

Control surface and control system parts shall be analyzed in neutral setting as parts of the remaining airplane and must withstand stresses, caused by displacement, with the same load factor as for the wings, but only for control forces of 30 kg on the arms, and of 50 kg on the legs; the factor of safety is 2.

The landing gear in normal flight attitude shall be designed to withstand 5 times the gross weight of the airplane. The force direction is 1) vertical, 2) inclined 10° forward, with the resultant passing through the fuselage aft of the vertical, 3) inclined at 10° forward and 15° toward the side.

In three-point landing the landing gear and the skid shall absorb the same load in relation to their support reactions in rest position. The heights of vertical drop of the airplane and the one-wheel landing are the same as in 1927. The remaining parts of the airplane shall be designed for a dynamic load factor of 6 without exceeding the elasticity (yield) limit. The case of nosing over at low speed shall also be analyzed.

These new regulations of the Bureau Veritas constitute a real step forward. They are logically carried through on the premises that the wing stress is linearly dependent on the control forces - an assumption which holds fairly true for not unduly rapid control operation and unbalanced or slightly balanced elevators. In addition, it is assumed that the pilot does not exceed stated stick forces, with the result that the horizontal boundary lines of the load factor and the hyperbolic boundary lines for the dynamic pressure are as shown in figure 42.

The analysis of cases B and C is contingent upon the highest attainable speed for which the exact directions are given under which the speed limitation of normal airplanes by control pressure is noteworthy, thus voiding the contradictions contained in the old B case with standardized lift coefficient $c_{a_B} \sim 0.3$.

Higher stresses than those given within ranges C and B (see fig. 42) are not possible, at least with airplanes without speed limitation (dive with terminal velocity permissible). It is more doubtful for airplanes without speed limitation (curve o, fig. 42) as to whether the permissible speed can always be maintained and whether the factor of safety against maximum possible stress remains $\geqslant 2$.

The greater expectancy of stresses within this range makes it necessary to figure with a strength lower than the tearing strength, i.e., to roughly divide the tearing strength by a factor of safety.

Higher stresses are always possible in ranges B to A when the permissible control is exceeded. And no pilot can guarantee to keep within a stated control force, especially not in a moment of danger. To this zone B-A the concept of safety conventional in structural statics against highest possible stresses also cannot be applied, for if applied, it would lead to such high breaking-load factors that an economical airplane design would become impossible. Since the loading condition A, is the primary consideration for the strength of the wing-truss structure and the least safety can be guaranteed for it in the usual sense of the word, the disinclination heretofore in airplane design against the factor of safety, will be readily understood.

A noteworthy feature is the demand to investigate gust stresses in cruising flight, as earlier advocated by Breguet. Despite the fact that the dominating effect of speed on the stress of airplanes was duly recognized again, one still could not make up his mind to give up the constant load factors, so expedient for static analysis.

9. Italian Loading Conditions

In his attempt at mechanically similar interpretation of the French formula

$$n_A = k \frac{F}{N} (0.036\ v_h)^3 = 0.007 \frac{k\ \eta}{\rho_o\ c_{w_h}}$$

for the load factor of case A, Rota (reference 66) investigated the relationship between wing weight, total weight of airplane, wing area, engine power, and speed for a number of airplanes. To be sure, the results obtained were not uniform because the discrepancies between the existing types are even outwardly too glaring.

The Italian specifications for breaking-strength tests (reference 67) as published in bulletin no. 13, are much like the French Regulations of 1923. But the load factor is not given according to formula but graded according to flight speed and gross weight of airplane.

In case A the resultant of the air loads is at one third of the wing chord and perpendicular to the plane of the wing. The breaking-load factors n_a are tabulated in

Table XXXII. Italian Load Factors, 1923

			G (t)		
V_h (km/h)	< 0.75	0.75 to 1.2	1.2 to 2	2 to 6	6
100	6	6	6	6	6
125	7	6.5	6.5	6	6
150	8	7.5	7.5	6.5	6
175	9	8.5	8	7	6.5
200	10	9	8.5	7.5	6.5
225	11	10	9	8	7
250	12	10.5	9.5	8.5	
275	12.5	11	10	9	
300	13	12			

(km/h × .62137 = mi./hr.)

$n_a' = 1.15\ n_a$ for acrobatic airplanes,

$n_a'' = 0.90\ n_a$ " military "

In case B the inclination of the resultant against the chord is $9\ n_a : 4\ n_b$. In case C, the wings are stressed by their own drag. The load factor in cases C and B is $n_b = n_c = 4.5$ for airplanes with identical or similar front and rear spars, and = 3.0 for all others. For inverted flight the ultimate load factor is ordinarily n_d' = 2 and for landing $n_d'' = 9$, wherein the most unfavorable case shall be decisive.

These regulations were revised in 1924 (reference 68). The load factors n_a for large airplanes were lowered considerably. (See table XXXIII.)

Table XXXIII. Italian Load Factors, 1924

V_h (km/h)	G (t)				
	0.5 to 1	1 to 2	2 to 3	3 to 4.5	4.5 to 6
500 to 400	15	13.5	12	10.5	9
400 " 300	13	11.5	10.5	9	8
300 " 250	11	10	9	8	7
250 " 200	9.5	9	8	7	6
200 " 150	8	7.5	6.5	6	5
150 " 50	6.5	6	5.5	5	4

The loading conditions for wing strength shall be proved by analysis or load test for

a) pull-out from a vertical dive, breaking-load factor n_a;

b) additive torsion by aileron displacement, speed as in case a (only for monoplanes and specially designed multiplanes);

c) vertical dive with terminal velocity;

d) inverted flight and landing.

The load factor in case a for wing elasticity tests is

$$n = \frac{n_a}{3} \geqslant 2.5.$$

To allow for unsymmetrical stresses, the breaking-load factor on one half of the wing shall be assumed as reduced by 1.

The load distribution over the span is proportional to the wing chord, but the wing tip at distance $t/4$ shall be assumed as unloaded. The ribs shall be investigated for triangular load distribution, once with the maximum over the leading edge of the wing, then at one fourth of the wing chord.

The upward and downward loading of the horizontal tail surfaces shall be assumed equal to the ultimate wing load n_a p. The up-load shall be in combination with three times the force produced by the static moment equilibrium (c.p. at one third wing chord). The loading of the vertical tail surfaces is $\frac{p\, n_a}{2}$, that of ailerons 250 kg/m² (51.20 lb./sq.ft.), and shall always be uniformly distributed over the fixed and movable surfaces. The dynamic load factors for the landing gear are given in the following table.

Table XXXIV. Dynamic Load Factors of Landing Gear

	Elasticity test flying		Ultimate load flying	
	day	night	day	night
Normal wheel landing	3	3.5	5	6
Oblique landing, $\alpha = 27^{\circ}$	2	2.5	3	4

In 1931 there appeared a draft by the Technical Committee for the Royal Italian Army and Navy, which was patterned after the CINA regulations, while introducing the factor of safety 2 of the German loading conditions.

Table XXXV. Italian "Safe Load Factors" n_{as}, 1931

Gross wt., airplanes (t)	< 1	1 to 2	2 to 3	3 to 4	4 to 5	> 5
Stress category N (normal)	3.5	3.3	3.1	2.9	2.7	2.5
Stress category S (special)	2.5	2.4	2.3	2.2	2.1	2.0
Stress category A (acrobatic)	5.5	5.1	4.7	4.3	3.9	3.5

Wings and Cellule

1. Flight with maximum lift coefficient. "Safe" load factor n_{as} in accordance with table XXXV. In strength tests this load is the test load, whereas for wood designs, it is 0.8 n_{as}.

2. Horizontal flight at maximum speed. "Safe" load factor $n_{bs} = 0.75\, n_{as}$.

3. Flight at zero lift. The turning moment M, to be absorbed by the horizontal tail surfaces, is applied at the wings. It shall be

$$0.5\ Gt < M = 0.2\ n_{as}\ Gt < 0.75\ Gt \tag{53}$$

4. Flight with negative lift corresponding to inverted flight or flight in bumpy air (at maximum horizontal speed); safe load factor $n_{ds} = 0.5\ n_{as}$.

5. Rough landing. "Safe" load factor.

e = 3	for category	N,
e = 2.5	" "	S,
e = 4.5	" "	A.

To allow for unsymmetrical stresses, the "safe" load factors n_{as} and n_{bs} shall be reduced on one side of the wing by 0.5 for category N and S, and by 1.0 for category A. If no wind-tunnel data are available, the distribution as given in figure 43 shall be applied. The unit (breaking) load of the vertical tail surfaces shall be

$$p_s \geqslant 0.6\ n_a \frac{G}{F} \geqslant 3.6\ v_h \tag{54}$$

and

$$100 < p = 0.04\ v_h^2 < 300 \tag{55}$$

for the ailerons.

The following "safe" control forces shall be assumed:

50 kg	at stick vertical to axis,
25 "	each at rim of wheel tangential,
50 "	on each rudder bar,
75 "	on both rudder bars.

By dual control 75 percent of the separate forces shall be assumed.

Landing Gear

The energy absorption of the shock absorber shall correspond to the height of drop

$$0.3 < h = \frac{G^{1.5}}{F} 10^{-4} < 0.7 \tag{56}$$

whereby the shock-absorber leg shall not be compressed exceeding 0.75 h.

a) Landing with center of gravity perpendicular above the wheel axle. The inclination of the wing chord forward toward the horizon shall not exceed 10°. "Safe" dynamic load factor 2.5.

b) Landing in horizontal flight attitude. The resultant passes through the wheel axle and the center of gravity. "Safe" dynamic load factor 2.5.

c) Three-point landing. "Safe" dynamic load factor 2.5.

d) One wheel landing in horizontal flight attitude. The transverse axis slopes at 15° toward the horizon. "Safe" dynamic load factor 1.5.

10. Dutch Loading Conditions

The Dutch specifications for airplanes of May 28, 1924, followed the English very closely. Classes I and II, as well as load cases a to c are, in fact, identical with them.

Each part of the airplane shall be so designed that it neither breaks nor becomes excessively distorted nor elastically deformed. For analyzing the different structural components the airplane shall be assumed to be impressed by the following loads without other outside forces.

1. Wings. The following load cases shall be analyzed whose load distribution, multiplied by the load factor in table XXXVI, corresponds to the air loads of these cases:

   a) flight with extreme forward position of the upward resultant of the air load;

   b) flight with maximum speed v_h at ground level;

   c) dive with terminal velocity. The drag of the nonrotating propeller may be included. But the diving speed shall not be assumed greater than corresponds to the control force for the tail surface loading cited below.

   d) inverted flight. All controls are in neutral.

Table XXXVI. Dutch Load Factors, 1924

Weight (t)	Class I			Class II	
	≤ 2.3	4.5	≥ 13.5	≤ 1.35	≥ 4.5
Case a	5	4	4	7.5	6
" b	4	3.25	3	5.5	4.5
" c	1.25	1.25	1.25	1.5	1.5
" d	-	-	-	3	3

In addition, the wings shall be designed to withstand loads which are transmitted to them from other parts of the airplane. The airplane shall still be able to fly and remain steerable after failure of one wing fitting.

2. Tail surfaces. The loading of the fixed surfaces shall be uniform, that of the movable surfaces tapering to zero at the trailing edge. The maximum pressure on the fixed surfaces and the leading edge of the movable surfaces is

$$p = \frac{v_h^2}{16} \sim q_h \tag{57}$$

3. The landing gear shall withstand the following load cases without aerodynamic forces:

   a) landing on both wheels, thrust line horizontal;

   b) three-point landing.- stress with at least 4 times the gravitational forces;

   c) landing on one wheel, side load $P = G$.

In load cases a and b, a 1.15 times safety factor against energy of striking shall be proved, which occurs by the absorption of the sinking energy through the landing-gear shock absorber. The rate of sinking is:

$$w = 0.9 + 0.09\, v_l \tag{58}$$

4. The fuselage shall be designed to withstand the air loads on the wings and control surfaces as well as 1.1 times the forces cited under 3.

Subsequently, Holland became affiliated with the CINA and amended its loading conditions in some respects.

In the tabulation of the breaking-load factors of table XXXVII, it was assumed that the maximum load in flight does not exceed 50 percent of the breaking load, except by the fastest and most maneuverable airplanes.

Table XXXVII. Dutch Load Factors after Joining the CINA

Weight (t)	Class I $\leqslant 1$	Class I $\geqslant 5$	Class II $\leqslant 1$	Class II $\geqslant 5$
Case a	5	4	8	6
" b	4	3.25	6	5
" c	-	-	1.5	1.5
" d	-	-	4	3

Class I, intended for commercial aircraft, provides no case c or d. But then case b shall be analyzed for that resultant of the air loads which exists at 1.3 times the maximum horizontal speed.

In case d the air load is applied at the same point but inversely from that for case a. The maximum pressure on the control surfaces is

$$p = \frac{v_h^2}{24} \sim 0.67\ q_h > 75\ \text{kg/m}^2 \qquad (59)$$

The tail load is, like in the U.S. specifications, dangerously low.

On December 6, 1928, the Rijksstudiedienst voor de Luchtvaart (Royal Institute for Aeronautical Research) issued new Technical Requirements for Airworthiness (reference 69).

Proof of sufficient strength for divers attitudes in flight and on the ground shall be adduced. For these attitudes loads are assumed which are termed possible ("possible loading"), and which are arrived at by multiplying "normal loading" in the pertinent attitude by a load factor.

To insure sufficient safety, the structure shall be designed to withstand the design load, which is found from the "possible" loading by multiplication with a factor of safety S. This factor of safety is the product of several other factors (subfactors).

The first subfactor, 1.5, for normal cases, gives a safety with respect to errors in analysis, load distribution, wear, etc.

The amount of the other subfactors depends upon the importance of the particular part, on the fact as to whether the material or the load schedule is such that, after exceeding the permissible stress, the particular part breaks immediately or that a greater distortion precedes the failure (more brittle compared to pliable material, buckling versus tension failure, etc.), and on the test possibility of the part, when the airplane is in service condition. Besides, these subfactors give a guaranty against uncertainties in load distribution and stress analysis.

For principal parts such as wings, fuselage, and landing gear, the total factor of safety S, as sum of these subfactors, shall ordinarily not be less than 1.8 for cases in which it can be assumed that the structure does not fail even after exceeding the permissible loading, whereas for column load and for less pliable materials the total factor shall not be less than 2.0.

The size of the subfactors which together form the factor of safety S, shall be proposed by the applicant and, after discussion with him, determined.

The material stress produced as result of the design loading shall not exceed the permissible stress established for each material. This permissible stress of the material is that mean stress at which no great distortion occurs after unloading. As a general rule, the permissible stress of materials having a distinct yield limit shall lie at that very limit. For material parts exposed to vibrations or shock, the "safe" stress against this kind of loading shall be considered the permissible stress.

The loading conditions for airplanes are:

a) The airplane flies at a positive angle of attack such that the c.g. is in extreme forward position;

b) Maximum horizontal speed v_h at ground level;

c) Gliding flight, with the speed equal to k times the speed in case b. The speed factor k is given in table XXXVIII.

The gliding speed used in the analysis shall be stated in the airworthiness certificate, thus making it absolutely clear that this speed may not be exceeded without danger and must also be shown in the airplane for the information of the pilot.

d) Inverted flight. The c.g. position as in case a, but in the opposite direction.

For analyzing the strength of the attachment of the wings with the fuselage and the forces which they exert on the fuselage, the following assumptions shall be made:

1. The airplane is fixed at the wings and subjected to a combination of loads consisting of 0.67 of the loads of case a or b, with 0.50 of the moment which the prescribed force exerts upon the vertical tail surfaces about the center of gravity.

2. The airplane is fixed at the fuselage and 0.67 of cases a and b loading is applied with

   I. A rolling moment from loading one wing-half with a lift of 0.5 of the total weight of the airplane;

   II. A torque by twice the maximum propeller thrust from the propellers located on one side, outside of the median plane of the aircraft.

For extremely maneuverable and speedy airplanes, higher load factors n than given in table XXXVIII can be asked.

Table XXXVIII. Dutch Load Factors, 1928

Case		Class I				Class II			
		a	b	c	d	a	b	c	d
Possible load factor	n	2.5	1.9	1.0	0.5	3.75	2.75	1.0	2.0
Design ultimate load factor		5	3.8	2	1	7.5	5.5	2	4
Speed factor	k	-	-	1.3	-	-	-	1.5	-

These loadings shall, as much as possible, be distributed according to the aerodynamic properties. Torsional and flexural stiffness shall also be taken into account in the dimensional analysis.

Horizontal and vertical tail surfaces shall be designed to withstand the mean unit loading

$$p = S \frac{v_h^2}{k} \geqslant 75 \text{ kg/m}^2 \tag{60}$$

whereby $k = 32$ for commercial airplanes, $k = 32$ to 48 for other airplanes with $v_h = 50$ to 100 m/s, and $S = 2$ to 2.2 as the factor of safety.

The horizontal surfaces shall be designed to withstand the maximum horizontal moments in cases a, b, c, and d.

The pressure distribution shall be assumed triangular and rectangular, tapering to one third over the elevator. Balancing surfaces must be strong enough to carry twice the pressure of the other surfaces.

The landing gear shall be analyzed for

1. wheel landing, thrust line horizontal, resultant through the center of gravity;

2. three-point landing with vertical reactions, which shall be at least twice the static load;

3. wheel landing with 0.67 of the vertical component obtained under 1, and 1:4 side load.

The energy absorption of the shock absorber shall, with 1.1 times safety, suffice for the same rate of sinking as in 1924.

For analyzing the fuselage from the control surface loading it shall be assumed that the fuselage is solidly suspended from the wing fittings and subjected to the greatest possible loading on the horizontal tail surface and to 0.33 of the load on the vertical tail surfaces.

The minimum factor of safety for the fuselage analysis shall be $S = 1.8$ to 2, and $S = 2.5$ for the engine nacelles. For fuselage with engine built in, a minimum $S = 2.5$ may be required for the engine boarers.

These loading conditions constitute a valuable contribution to the safety problem. The rule specifying that the yield limit shall not be materially exceeded even by breaking load, is especially noteworthy. Loads up to near the breaking loads can be readily sustained by such designed parts without affording appreciable distortion or internal injury. This quality, which is attainable for tension members by a slight increase in weight, may be considered as a well-worth-while aim of light-structure design.

The design schedule for fuselages may seem slightly amusing, but it may conform in simple fashion to experience.

11. Russian Loading Conditions

The regulations established by the Central Aerohydrodynamic Institute of Moscow, and patterned after the German regulations (reference 70), went into effect August 1, 1927.

The stresses in flight (safe loads) shall be analyzed experimentally or theoretically. The stipulation for the stress analysis is simply the product: ultimate load factor = safe load factor times factor of safety. The following cases shall be analyzed (ultimate load factors are given in table XXXIX). The load cases for the wings are:

Case A: by maximum lift coefficient, resultant inclined at 98° to the wing chord;

Case B: exactly as that of the BLV, 1916;

" C: dive with terminal velocity. Torsion moment and frontal resistance on the wing shall be analyzed according to wind-tunnel data. The formula for propeller drag in diving is

$$W \sim \frac{\rho v^2}{2} F_{sA} \left(1.38 - 0.63 \frac{H}{D}\right) \tag{61}$$

The propeller pitch $0.5 \leq \frac{H}{D} \leq 0.9$ shall be measured by 0.7 outside radius.

Case D: exactly as in the BLV, 1916;

" E: wing stress by landing impact. (See farther on.)

The wing ribs shall be investigated for the load distribution given in table XLIV.

The load distribution across the span is given for six conventional wing sections. The wing loading tapers at the tips from 0.5 or 1.0 mean wing chord to half.

The ailerons shall be designed for a mean ultimate load of $p = 0.0525\ v_h^2 \geq 125\ kg/m^2$. The loading forward of the axis of rotation is uniformly distributed with linear drop to one third along the chord. The horizontal tail surfaces shall be analyzed for the stresses in case C with a load factor n_{cH} higher than that of the wings. They shall also be strong enough to withstand the ultimate load

$$P = 0.196\ F_H\ v_l^2 \tag{62}$$

The same applies to the vertical tail surfaces, except that the coefficient is n_{cS} instead of 0.196. (See table XXXIX, page 29.)

The landing-gear analysis shall include:

three-point landing,	$2 + 0.18\ v_l > 6$
side load on both wheels,	$0.036\ v_l > 0.8$
shock from front parallel to thrust line,	4

Conformably, float supports shall be analyzed with the dynamic load factors:

$3 + 0.18\ v_l > 7,$ stern and bow impact

$-1 + 0.18\ v_l > 3,$ side load (inclination 1:4)

4, shock from front, parallel to thrust line.

The fuselage shall be designed to withstand the stresses in flight and landing; in case A with increased safety (ultimate load factor n_{aR}). A lateral ultimate load applied at the nose of the fuselage of from 3 to 4 times the weight of the forward part of the fuselage, is demanded.

12. Tendencies toward Uniform and Representative Formulation of Strength Specifications for Airplanes

This survey has revealed a confusing abundance of regulations which an airplane must comply with in order to receive official approval.

The underlying principle of these specifications - the load factor - reaches back to the beginning of flying. Originally this term defined only the strength of existing airplanes. During the course of development, especially from experience on a great number of airplanes of the same kind during the war, the load factor was given a reality purport which did not stand the test of subsequent experience to the extent anticipated. To illustrate: For all commercial airplanes of the same weight, or for all training airplanes, one definite load factor was thought sufficient to avoid failures in flight. This statement needs to be qualified, however; that is, in such airplanes only the probability of failure may be small. To make this assertion with a positiveness that would be equivalent to an absolute truth is unsubstantiated because the experiences which finally led to specifying the definite load factor, were themselves confined to only a limited number of airplanes within a limited span of time, and for that reason are simply utterances of probability.

Table XXXIX. Russian Load Factors, 1927

Weight G (t)	Commercial Airplanes				Military Airplanes							
	< 2.5	2.5-5	5-10	> 10	Bomber	Torpedo, observation	Water, observation training	Land, observation training	Water, two-seat	Land, two-seat	Water, single seat	Land, single seat
Case A	5.5	5.5-5	5-4	4	5	6	7	8	9	10	11	12
Case B	4	4-3.5	3.5-3	3	3.25	4	4.5	5	5.5	6	6.5	7
Case C	1.25	1.25	1.25	-	1.25	1.4	1.5	1.7	1.75	1.8	1.9	2.0
Case D	-	-	-	-	-	2	2.5	3	3.25	3.5	3.75	4.0
Case n_{aR}	7.5	7.5-7	7-6	6	7	8	9	10	11	12	13	14
Case n_{cH}	1.5	1.4	1.35	-	1.3	1.5	1.7	1.9	2.0	2.1	2.15	2.25
Case n_{cS}	0.196	0.196	0.196	0.196	0.196	0.245	0.245	0.294	0.294	0.294	0.294	0.294

(t x 2204.62 = lb.)

Indisputably the stresses of an airplane are physiologically and psychologically profoundly influenced by the pilot of the airplane. Because this relationship is difficult to express in figures, it was similarly concluded that the other physical influences on the stress which occur regardless of the pilot, did not have to be investigated any more closely, with the result that the whole experience was summed up in figures - the ultimate load factors - which were graded to conform to the different purposes of use. The advantage of simple specifications was thereby obtained at the price of lack in adaptability to the technical advances made in airplane design. The loading conditions in consequence had to be amended periodically, whereby profound sagacity was used to cast the specifications into more or less perfect yet simple form. One main purpose of this report was to preserve the many valuable recommendations and suggestions in this respect hitherto proffered.

When applying the specified load factors to new, more powerful airplane types, a number of failures occurred which no longer could be reconciled with the collected experiences, and made tightening up of the regulations imperative. At the same time these accidents raised the question of the underlying principles of the strength specifications as a whole, because of the danger of repetition involved, unless the physical cause of the stress is analyzed.

The loading conditions of the various countries show, even to-day, a wide divergence from one another, a case in point being the ultimate load factor in case A, illustrated in figure 45. Owing to its international aspect, standardized design requirements, valid for a considerable period of time, are urgently needed. This desire likewise found expression during the First International Safety Congress, in Paris, December 1930.

During its session there, the Committee on Airplane Structures recommended, with due regard to the aims of the CINA, to increase the safety on the international air lines by greater structural strength of the airplanes and deplored the absence of uniform design specifications in the different countries. Complete unification of these specifications should be aimed at

1. in the methods for the determination of the maximum forces which affect the individual parts of the airplane structure;

2. in the load factors for the different flight cases and their application to the different parts of the airplane;

3. in the methods used for determining the inside loads set up by the cited outside loads;

4. in the assumed practical values for the mechanical property of the structural materials (ultimate strength, apparent and proportional elastic limits, fatigue strength);

5. in the factors of safety, i.e., the relation between the actual breaking strength of a member and the maximum possible stress which this structural component has to absorb.

At the same time, the Commission for Organization and Statistics recommended that

1. every country should publish official accident reports for civil aircraft and to specially stress the much smaller risk in commercial aviation (reference 71);

2. accident statistics be standardized, since the progress in aviation demands the systematic study of accidents as an essential basis.

This worthy aim of unification of strength specifications appears, however, according to all previous experiences, to be attainable only when the multiplicity of aeronautical problems are more taken into account than hitherto, and when these problems (stress analyses) are carried out on a large scale along internationally agreed lines.

The existing specifications for airplanes relieve the designer of an essential share of his responsibility and give him a not always causative feeling of safety.

The ideal state in airplane design - realizable, perhaps, in the remote future - is complete freedom and responsibility of the designer for the choice of sufficient strength. But this state presupposes a very reliable and widely diffused knowledge of stresses and a certain interruption to technical development. At present we are far from that stage. There is some justification in calling present-day air transportation experimental operation.

So long as these presumptions are not fulfilled, it would hardly be wise to discard minimum requirements for the strength of airplanes. But those requirements should be so formulated as to vest as much responsibility as possible in the designer rather than to tie him down to specific load cases. From the technical point of view, it would necessitate bringing the physical process of the stress closer to the designer by suitably formulated regulations and by supplying him with research data regarding the anticipated expectancy of the stresses. This would enable the designer to analyze the stresses of the airplane himself from certain prescribed initial conditions and in that way to take into account the more or less propitious characteristics of his design project. Ostensibly, this method is so superior to that of the orthodox schematic coefficients that the increased paper work involved is of no consequence.

When, at present, it takes about 200 working hours to merely analyze the stresses of the control surfaces, the labor of 50 hours more for computing the outside loads acting on the control surfaces is not prohibitive, because the 200-hour static calculation is not made for its own sake but rather to assume a substantially safe knowledge of the outside loads. To illustrate: If approximation methods with the least total error were used for the stress and the aerodynamic analysis, one might perhaps become of the opinion that it takes 200 hours to calculate the outside loads, inclusive of a test in the wind tunnel, and but 50 hours for the stress analysis by approximation method. This illustration is typical of many other cases. The accuracy of the stress analysis in airplane design today is still in marked disagreement with the accuracy of our knowledge on air loads. And since this knowledge cannot be increased at once, but only in slow stages by wearisome experimental work, a lesser degree of accuracy in static analysis may be permitted for the present if thereby the designer is relieved for the more exact investigation of the outside loads. Should the designer be averse to undertaking comprehensive investigations into the outside forces and making detailed stress analyses - perhaps because the smallness of the economic object prohibits this - a condensed stress analysis with loading conditions simplified toward the safe side, might be permissible.

Unquestionably there will be much opposition at first on the part of airplane builders, against any change in design practice, but the change is necessary and will have to come sooner or later.

When discussing the causes of accidents, which really have supplied up to now the chief reason for changing the strength specifications, the designer is wont to ascribe the fault to the pilot, unless plain material or manufacturing defects have been proved, and to stress the fact that no designer could design a foolproof airplane; that flying is, and always will be a dangerous profession, and is against any and all tightening up of specifications which would lower the useful load. This stand is justified to a certain extent. A careful and skilled pilot can fly even a less strong airplane in safety, provided he very scrupulously refrains from high speed, flying through clouds, flight at low altitude, rapid control maneuvers, etc. One weighty argument on the part of the designer quite often is the assertion that an experienced pilot has the right feel for acceleration and that the tradition of artistic flying itself prohibits the exceeding of certain elevator-displacement speeds and accelerations.

But the fallacy thereof is proved by the accidents which do happen to very experienced flyers, and in which the airplane flying without useful load must have reached abnormally high accelerations at failure which exceeded the usual amount many times. It is therefore not advisable to depend on ideal pilot qualities.

Should it be undesirable to increase the hazards of air transportation, especially with new types of airplanes, it will then be necessary to assume very unfavorable, physically possible interactions of the pilot on the control system within a certain speed range with which the pilot is familiar and to include them in the analysis.

Lastly, as far as the main worry of the airplane designer - pay load and speed - is concerned, it should be remembered that this recommended change in design practice does not necessarily imply a greater design weight of the airplane.

The very fact that for each design the outside loads must be determined conformably to the prescribed initial conditions, offers new possibilities in weight saving by appropriate outer forms of the airplane and in eliminating indiscriminate, superfluous material accumulations in the airplane. These possibilities, however, should only be attempted in connection with the problem of flight qualities. It would be erroneous to sacrifice good flight qualities in order to insure low stresses. Highly loaded airplanes in particular, promise considerable saving in weight in this respect, without increasing the probability of accident. How much actually is attainable in this way has been proved by the record flights with overloaded airplanes which in part were made with extremely low ultimate load factor without incurring wing failure. A schematic application of these load factors in continuous service, on the other hand, may lead to fatigue failures. Thus it is seen that the type of the produced stresses and the stress procedure must be first analyzed in detail, before proceeding to the stress analysis.

The service life of a modern airplane is still quite short as compared to other vehicles. The number of loadings and unloadings of the wing is comparatively small, the load changes in flight are, in so far as frequency is concerned, of such low magnitude as to have only rarely induced fatigue failure. The same applies to control surfaces. The necessary resolution of airplane design into thin-walled components, postulates low specific material wear and less stress and fatigue failures than stability failures.

For the present at least, most severe failures in vitally important parts of an airplane, such as wings and control surfaces, are unquestionably caused by one-time effective, particularly great, outside loads. Aside from that, there are, of course, a greater number of fatigue failures on engine supports and on the body end and skid, but which as a rule are not serious. In those parts the number of stress reversals due to the inertia resistance of the gearing or shocks when dragging over rough ground, attains to the order of magnitude of 10^6 after a short time and thus may induce fatigue failure. As a rule, the number of fatigue failures in general machine construction, as well as in airplane engines, is admittedly greater than all other types of failure (reference 72).

As concerns increase in the average life of an airplane, the designer must be given data regarding the expectancy of rare, but extremely high stresses as well as about the anticipated expectancy of all low stresses which may occur in service.

These last-cited data will under certain circumstances serve less for the dimensioning of airplane parts than for computing the life span which the airplane can probably reach without fatigue failure. For it may be far more economical to replace an airplane after a stated period of service by a new one than to drag along the additional material quantities necessary to avoid fatigue failure as dead weight during the entire service period, in the face of an expectancy of perhaps 0.8 that the airplane becomes obsolescent or is lost through some cause or other before completing its service period.

Our knowledge on the expectancy of stresses in actual service is as yet so meager that statistical data are scarcely possible (reference 73). Such data are not to be expected for some time, until the results of the statistical work now undertaken, have been completed. They may be utilized in two ways.

First, it will be possible to effect the testing of airplane materials or built-up components for fatigue strength true to actuality, by permitting the amplitudes to increase or decrease during the test under assumption of uniform distribution over the test period periodically conformably to the expectancy curve measured in flight. This may be accomplished by electromagnetic fatigue-strength testing machines with grid-tube control (reference 74). This kind of fatigue strength has not been explored heretofore; neither has there been much research into fatigue strength in the narrower sense by constant amplitude in the range of smaller number of reversals of the order of magnitude of 10^1 to 10^5, because it has little significance for general machine design. But for airplane design, knowledge of this range in connection with the expectancy curves recorded in free flight, will be very valuable.

The more important practical result of the statistical research will presumably be that from these expectancy curves the probable expectation of failure in unit time is estimable for a one-time appearance of extraordinarily high outside loads by extrapolation in direction of lesser expectancy, say, conformably to Gauss' error curve and determination of the section point with the prevailing ultimate strength. (See fig. 1.) By such investigations the airplane crash data which up to now formed the basis of the strength requirements, can be extended in a less sacrificing manner. Aside from that, the crash data should be interpreted from careful inquiries of the accident and of the qualities of the crashed airplane as far as possible, and the statistical data should start with airplanes of that very type after satisfactory strengthening of the broken part, in order to measure the expectancy of the cause of failure or to determine the limits of error of the above-cited extrapolation. It is in this direction that the D.V.L. is actually proceeding and by means of which, strength requirements true to reality will be gradually evolved.

The data on failures collected so far, are partly incorporated in the present strength requirements and give even now some valuable hints for the future formulation of strength requirements, of which the following is a brief resume.

Stresses in Flight

Our accident statistics prove that all airplanes should be at least strong enough to withstand the stress produced by pull-out from maximum unaccelerated horizontal flight at ground level (fig. 46). This stress while rare is nevertheless not so improbable even with experienced pilots as to merit no consideration

In sharp pull-out, normal force coefficients are temporarily possible which far exceed the maximum normal force coefficients recorded in a steady attitude, because the separation of the unsteady flow as result of lack of time to form a dead air space does not occur except at high angles of attack (reference 75). The height of the normal force coefficient reached by pull-out is dependent upon the strength of the pilot, the type and dimensions of the horizontal tail surface, and the retarding effect of the damping members parallel to the elevator. The stresses in small maneuverable airplanes are usually higher (reference 76).

In order to allow for all these effects it appears expedient to begin the investigation with a stick force of about 40 kg (88.18 lb.), which corresponds to two-handed continuous pulling and pushing on the control column (reference 41*). If automatic control systems are used, the available maximum pressure for manipulating the servo piston could be used as basis for analyzing the initial stress (reference 77).

It is extremely difficult to determine the stress of slotted wings with automatic flap. Abnormally high normal forces are possible which await determination by experiment.

In small airplanes a 40 kg stick force may suffice to put the elevator quickly hard over. But in large airplanes it would be quite difficult to analyze the elevator displacements in that manner because the elevator moments are markedly affected by small variations of the balance. Here is where statistical research must supply the information.

The physiological fact that the type of control operation is decisive for the stresses, is not to be denied in spite of it.

As against cruising speed the maximum horizontal speed merits the preference as basis of the investigation because in most cases it forms the limit of the speed range with which the pilot is familiar and which can be reliably established. In special cases, where a large power reserve is used only for starting, climbing flight or for saving the engine, the above cited requirement may be used when the pilot guarantees to keep within the narrower speed limit. But even in that case, the 1.2 times cruising speed at the lowest will have to be used as basis because the failure happens only by the coincidence of two, in themselves, rare results, namely, exceeding the usual cruising speed and applying more than the customary stick force.

In heavily loaded airplanes, i.e., such as even by full horsepower can fly horizontally only with high-lift coefficient, the attainment of maximum normal force coefficients is more frequently to be expected, especially when the gust stress is included. The analysis of airplanes of that kind should therefore include the fatigue strength of the material and the investigation of sharp pull-out can likewise be extended to include higher speeds.

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*For references 39 to 60, inclusive, see Part II (T.M. No. 717).

In the existing strength requirements the stresses at high speed in vertical glide and dive take up considerable space. This is due in part to the classic two-spar wing design, with its low torsional stiffness, in which high strength was required so as to insure sufficient rigidity in service, and in part to the marked center of pressure travel of the old wing sections. At present a large percentage of all airplanes - commercial airplanes in particular - rarely, if ever exceeds its maximum horizontal speed, and then not very materially. Besides, these airplanes often have, for structural reasons, very torsion-resistant wings and wing sections with fixed center of pressure. Their strength is therefore largely contingent upon the pull-out from maximum horizontal flight and the stresses produced in gusts. Aside from this, it is advisable to limit the maximum gliding speed for all airplanes to a push of about 40 kg on the control column insofar as the special purpose does not call for protracted gliding flights with a certain gliding angle. In small airplanes the thus-characterized gliding speed will correspond to the terminal dynamic pressure in diving, that is, the maximum attainable dynamic pressure, whereas in larger airplanes the gliding speeds may be considerably lower, depending on the type of design.

When the special purpose of an airplane is other than frequent diving or gliding at high speed and does not call for special fatigue tests of the material, the airplane can be designed so as to withstand the maximum gliding speed as well as the subsequent process of pull-out.

The pull-out can be visualized as the pilot releasing the control and the airplane by virtue of its longitudinal stability tending toward a greater angle of attack. Hereby, without appreciable change in flight speed, the ambit of small angles of attack is rapidly passed, during which the normal force grows proportionally to the angle of attack. The maximum normal force finally acting on the wing, depends on the shape of the control surfaces and should at least correspond to the lift coefficient $0.25\ c_{a\ max}$ (reference 76). Within range of higher angles of attack and lower flight speed, a linear course of the normal forces up to that of sharp pull-out from maximum horizontal flight can be assumed for the time being. (See fig. 47.) Simpler yet is the assumption of a constant normal force in this angle of attack range (fig. 42 and the dotted line in fig. 47), which should suffice for wing sections with little center of pressure travel. A recent suggestion (reference 84) yields load factors inversely proportional to the gliding angle by an assumed constant total drag.

Some elucidation on this still probelmatical angle of attack range by statistical research is very much desired for the future. For special-purpose airplanes - for stunt flying, for instance - the assumption of sharp pull-out must be extended to include higher gliding speeds also. In the extreme case, the maximum gliding flight, dynamic pressure alone would then be decisive for the strength (intersection of dot-dash lines in fig. 47).

During rapid change from horizontal flight to steep glide, as well as in certain other flight evolutions, the normal force acts in opposite direction on the wing, so that such stresses should likewise be investigated in training and acrobatic airplanes, the analysis again beginning with the maximum horizontal speed and a push on the control stick. The wing stresses in inverse direction due to gusts are treated elsewhere. In a controlled roll the tail surfaces are subjected to considerable torsional stresses about the longitudinal axis of the airplane.

Aside from these, the analysis of the stresses in flight evolutions appears unnecessary, because they either are smaller than the initial stresses or else the evolutions, such as looping, consist of pull-out motions.

An exception is the rudder and the aileron control. Here again one proceeds from the maximum horizontal speed and with stated continuous stick forces, say about 70 kg (154.32 lb.) foot power for rudder control, and a stick force of about 10 kg (22.05 lb.), or a moment of about 15 kg (33.07 lb.) for aileron control. The maximum temporarily executable stick forces are substantially higher, and run as high as 160 kg (352.74 lb.) for the elevator, 275 kg (606.27 lb.) for the rudder, about 45 kg (99.21 lb.) for the aileron, or 35 kg (77.16 lb.) as couple (reference 41). But these more than four-times-higher forces have to sustain only the control surfaces - at the most, the movable surfaces - because their effect is only intermittent.

The highest attainable rate of displacement for all controls and all control forces is about 2 m/s (6.56 ft./sec.). For small and medium-sized airplanes, it may be assumed that the continuous control forces themselves are sufficient to move the controls hard over within 0.1 s.

This assumption facilitates the determination of the stress in analysis (reference 78) or wind-tunnel experiment.

Simplifications such as these toward the safe side are desirable in limited number for the design of small airplanes, because in these many pieces must be made of heavier size anyway, for various reasons, and the paper work incidental to stress and strength analysis is usually not justified by the little gain in useful load.

Apart from the arbitrarily produced stresses, those induced by irregular atmospheric disturbances (gusts) must also be analyzed, especially insofar as it concerns commercial airplanes. The basis of the analysis is again the maximum horizontal speed at ground level.

The intensity of the stresses depends upon the speed of the unsteady air currents in direction, extent, and type of transition into undisturbed atmosphere. These influences can be mathematically segregated and in particular, it is possible to correlate the influence of the oscillation frequency of the wing with the stress intensity (reference 79). But the interpretation of the flight measurements as well as of the strength requirements postulates first an idealization consisting of assumed wing rigidity and simultaneous entry of both halves of the wing into an extensive air current at right angles to the direction of flight, whose velocity over a very small distance rises from zero to a constant value w.

This velocity is quite often $w = 10$ m/s (32.81 ft./sec.); more seldom $w = 13$ to 15 m/s (42.65 to 49.21 ft./sec.) in Central Europe, according to flight tests and accident statistics. It is more dangerous for small wings amd tail surfaces because they are more quickly engulfed within a dangerous zone than large surfaces. For the present at least, all airplanes should be so designed as to withstand an ideal gust of $w \sim 20$ m/s (65.62 ft./sec.), for one time, and in special cases the wing resistance against fatigue failure should be investigated conformably to the expectancy of the lower gust stresses determined from flight tests. These figures are subject to climatic conditions. The exploration of the expectancy curves necessitates statistical research in different climatic zones.

The problem of partial superposition of gust stress and intentional stress must for the time being be deferred, in spite of its importance. When, later on, individual expectancy curves for both types of stresses are available, then the height of the probable superpositions can be estimated.

The distribution of the air pressure over the wings and tail surfaces must be carefully analyzed in all cases. In load-carrying wings the stress is decisively affected by this distribution.

Considerable data are available on this subject (reference 80) which, although it applies only to steady flow conditions, should suffice provisionally.

The problem of wing and tail flutter should also be made the subject of a special analysis. In order to avoid this flutter within the whole speed range, it is necessary to insure a certain minimum degree of stiffness and in the vibration frequencies of the wing, the prediction of which is extremely difficult. The difficulty lies less in the investigation of the flow processes on the oscillating wing than in the analysis of the static oscillation frequencies of such a complex structure as is the airplane. Excepting the cantilever monoplane, the analysis will for the present at least have to be limited to the experimental determination of the static oscillation frequencies on the complete airplane, followed by an investigation to determine whether or not forced wing or tail vibrations are possible within the particular speed range.

The dangerous wing flutter observed up to now in flight as well as the accidents caused by wing flutter, were primarily due to aileron oscillations. Two known commercial high-wing monoplanes developed severe wing flutter at cruising speed when the aileron control cables became loose. According to various flutter investigations, observations in flight and accidents, the ratio of wing chord to wave length of an oscillation which gives a measure of the intensity of the required oscillation frequency of the wing, ranges between 0.20 and 0.38. The exciting conditions are ostensibly most propitious in this range. There are no oscillations when this ratio is greater, because the decreased energy absorption no longer suffices to overcome the ever-present material damping. The ratio does not exceed 0.27 when the air and mass forces of the ailerons are carefully balanced. Any further feasible wing vibration with the two degrees of freedom - bending and torsion -

are more easily and reliably computable and structurally controllable (reference 48).

Wing flutter can be avoided when the designer gives the wing such a high torsional vibration frequency (and through it, rigidity) that at the critical speed stipulated in the strength specifications the ratio of wing chord to wave length is > 0.38 or > 0.27 by careful aileron-mass balance. As to tail flutter caused by eddy separation on wings, on the fuselage, or propeller blades, it should be borne in mind that the frequency of the vortex streets formed aft of these parts is proportional to the flight speed, and does not coincide with the frequencies of the torsional vibration of the fuselage and the flexural vibrations of the tail, and that as far as possible all integral relations of these frequencies be avoided (reference 81). The rear end of the fuselage in particular, should have ample torsional stiffness.

Stresses at Take-Off and Landing

The stresses at take-off from level ground or calm water are reliably computable, since the accelerating thrust of the propellers or of the starting catapult is accurately known. When the ground is other than level, or the water is not still, stresses occur during taxying which heretofore were not amenable to analysis, because the ground obstacles, bunches of grass, furrows, snowdrifts, and the form of the water waves are so diversified that statistical data alone can give a solution. These taxying stresses moreover depend upon the speed and on the shock absorber, respectively, the form of the float-support system. The highest stresses ordinarily occur with a medium taxying speed, where the weight of the airplane is for the most part carried by the wheels or the float supports, respectively. These stresses occur so frequently that investigations of the stressed parts for fatigue are necessary conformably to the expectancy curves.

The most elementary kind of landing is the steady glide with throttled engine and minimum sinking speed. Even though the pilot usually flattens out over the ground to soften the landing shock, the above-cited method of landing is frequently used and is, in fact, necessary when landing in the dark or at places surrounded by obstacles. In order to allow for local irregularities of the landing area or slope of the wave crests and their velocity, the gliding angle for slowest gliding flight with power off as well as the inclination of the lateral axis should be suitably complemented in accord with the surface conditions of the ground, the seaway, and the wind (reference 82). One must also figure with the possibility that the landing is not exactly into the wind; that is, that the airplane has a lateral drift.

The vertical drop of the airplane to the ground can be effected at different inclinations of the longitudinal and lateral axis, which can be limited to the following positions:

1. Thrust line horizontal;

2. Flight attitude with maximum angle of attack at the limit of longitudinal stability;

3. Inclination of lateral axis until the wing tip touches the ground.

Then the landing gear or floats are to be strong enough to withstand the most unfavorable combination of these loads. The landing gear shock absorber may be allowed to travel up to its stops, but the stops should never be reached at a low frequently occurring stress, or else the danger of breakage increases enormously. The remainder of the airplane, especially the passenger cabin or cockpits should be so designed as to withstand stresses which are from 10 to 20 percent higher.

The impact on landing gear or float supports can be analyzed conformably to the laws of eccentric impact of a free body against a rigid wall (reference 83), provided the load increment is known as function of the time and travel or depth of immersion. But since this is not obtainable in a great many cases except by admitting extensive simplifications (reference 45), the information must be gained from measurements of the normal and rotary accelerations incurred by the landing shock or by tension measurements on individual parts of the structure (reference 47).

In oblique landing the second impact upon contact on the other wheel may be more severe than on the first, and the skid impact following a wheel impact can likewise be more severe than the first skid impact in a tail landing. Aside from the damping effect of the wings and tail by landing impact the impact forces at the first and second contact are identical when the product of the distances of the shock normals from the center of gravity is equal to the square of the inertia radius.

In which case the strength of both wheels or floats is utilized to best advantage.

The stresses in miscarried or forced landing require special consideration. It has already been mentioned that the chassis should have a lower breaking strength than the other parts of the airplane, especially the passenger cabin or cockpits, if it affords the passengers protection against injuries. The nose of the fuselage or the engine mount should be able to sustain considerable distortion after failure of the chassis. The passenger cabins should sustain as little damage as possible when the nose of the fuselage breaks or the wing hits the ground. Since it is physically possible to stand instantaneous accelerations as high as 10 for a short period without harmful effect it is especially desired that the cabins including the seats and safety belts of small airplanes be designed to withstand a 10-time force of gravity in direction of the engine axis. For large airplanes the forced-landing expectancy is less (reference 71), and consequently, the ratio of distortion by failure of chasis, of tip of fuselage or wings to the total kinetic energy at landing is for such airplanes usually greater than in small aircraft, for which reason the damages in a forced landing are never as severe as with small airplanes.

One may therefore question whether the considerably greater weight of large airplanes, which would be necessary to obtain the above cited strength could not be used to much better advantage for strengthening other vitally important parts or for improving the instrument equipment, in order to bring the total accident expectancy to a minimum.

Injuries to passengers are rare in forced landings of seaplanes. The protection of passengers against drowning is a purely constructive safeguard, watertight compartments and large fore-body length for protection against capsizing.

In conclusion we mention the stresses due to propeller thrust and gyroscopic moment, due to application of wheel brakes and towing as well as the requirement, especially as concerns small airplanes that vital parts may not be endagered by handling.

Summary

The historical development of the rules for structural strength of aircraft in the leading countries is traced from the beginning of flight to date.

The term "factor of safety" is critically analyzed; its replacement by probability considerations has been considered desirable.

One feature common to the strength specifications of every country is the application of the actual strength of existing airplanes of approved types primarily proportional to the total airplane wieght or the aerodynamic surface to new designs. The studied disregard - due to this simplification - of the more or less complicated process of stress in the face of the consistent advance in aerodynamics could not fail to show as result that efficient new airplane types were impressed by greater stresses than the experiences collected conformably to the older types led one to anticipate. The consequence was the need for repeated changes in the strength specifications.

In order to prevent the technical development from overtaking the specifications as much as possible it has been deemed necessary to analyze the physical process of the stress more in detail than hitherto with the aim toward a truly representative stress analysis, for which various suggestions and recommendations are outlined.

Translation by J. Vanier,
National Advisory Committee
for Aeronautics.

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Key to Abbreviations

ARC : Aeronautical Research Committee (London); until 1920, Advisory Committee for Aeronautics.

AVA : Aerodynamische Versuchsanstalt Göttingen (Aerodynamic Laboratory, Göttingen).

DLA : Deutscher Luftfahrzeug-Ausschuss (German Aviation Committee).

DLH : Deutsche Luft Hansa A.G. (German Luft Hansa).

NfL : Nachrichten fur Luftfahrer (Notice to flyers).

STAe: Service Technique de l'Aeronautique (Technical Service of Aeronautics).

WGL : Wissenschaftliche Gesellschaft für Luftfahrt (Scientific Society for Aeronautics).

ZFM : Zeitschrift für Flugtechnik und Motorluftschiffahrt.

NACA: National Advisory Committee for Aeronautics.

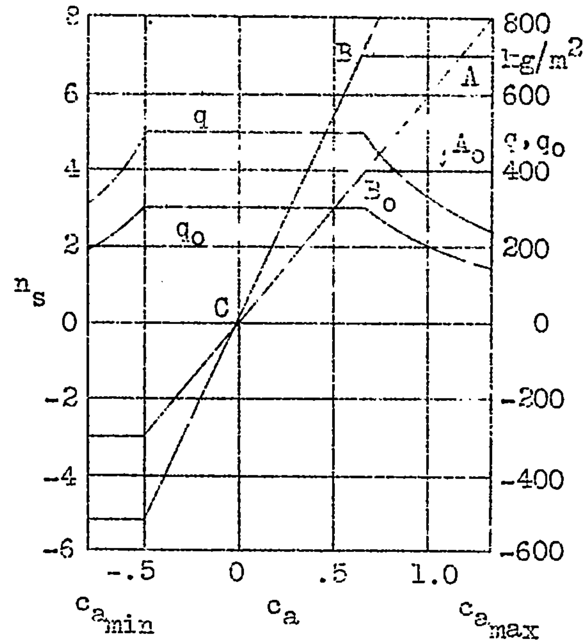


Figure 42.-Safe load factor n_s by dynamic pressure q for two acrobatic airplanes (index 0. with speed limitation).

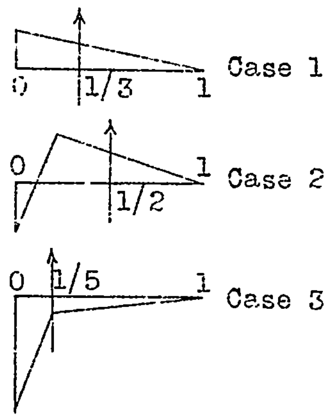
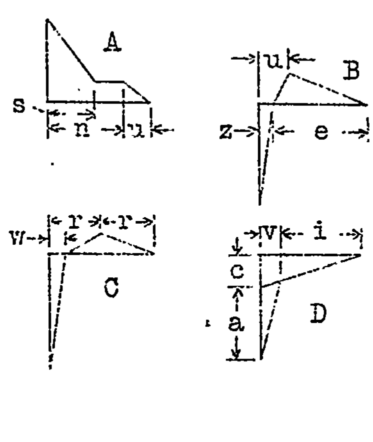
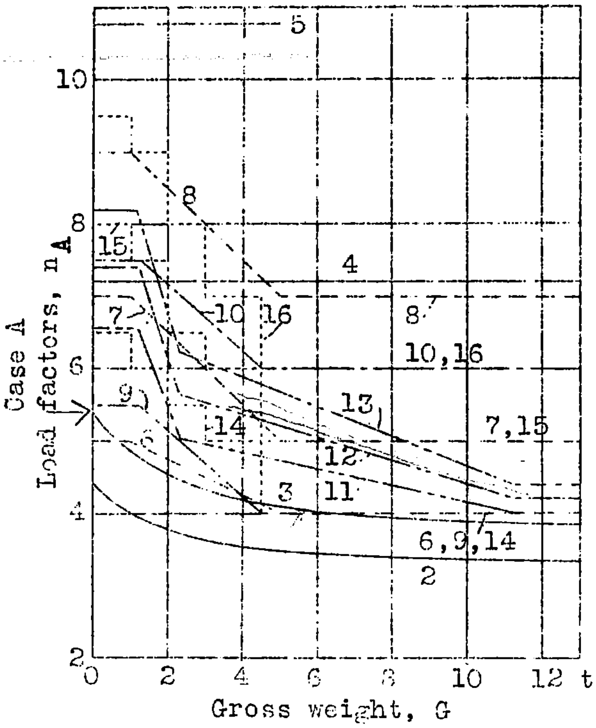


Figure 43.-Load distribution in direction of wing chord (Italy suggestion, 1931).



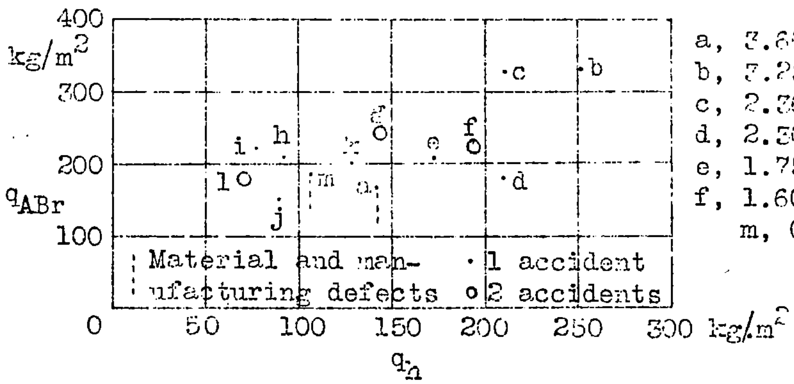
a, 2,275 H	s, .45 l
c, H	u, .25 l
e, .9l	v, .2 l
i, .8 l	w, .15 l
n, .75 l	z, .1 l
r, .5l	

Figure 44.-Load distribution over wing ribs. (source: loading conditions of the Soviet Ud SSR, Figs. 18 - 21).



Germany. line 2 to 5 ——— stress categories 2 - 5.
CINA. line 6 to 8 — — — class special, normal and acrobatic.
England. line 9 to 10 —·—— normal and acrobatic.
U.S.A. line 11 to 13 —·—·— power loading 9, 6 and 3 kg/hp
Italy line 14 to 16 ------ speed limits 150, 200 and 250 km/h

Figure 45.-Comparison of ultimate load factors in case A as of 1929.



a, 3.60 g, 1.30
b, 3.23 h, 0.84
c, 2.36 i, 0.80
d, 2.30 j, 0.63
e, 1.75 k, 0.57
f, 1.60 l, 0.45
m, 0.54

Figure 46.-Wing failures. Dynamic pressure at failure q_A computed from the highest steady lift coefficient. The figures denote the gross weight of the airplane in t.

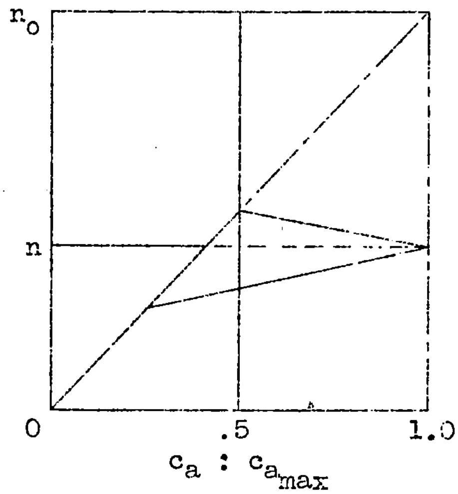


Figure 47.-Wing stress versus lift coefficient.